DESIGN & OFF-DESIGN PERFORMANCE ANALYSIS OF AN AERO ENGINE

by

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ABSTRACT : A Jet engine model has been developed to simulate its performance by matching the components using an iterative procedure. An analysis of the effect of variation in component efficiencies and specific heats on the performance of a twin-spool turbojet engine is presented for a set of flight Mach numbers and flight altitudes. A sensitivity analysis of specific thrust and specific fuel consumption indicate that there is a predominant effect of component efficiencies on engine performance.

1. INTRODUCTION

With a boom in air travel in the last two decades the performance of an aero engine has gained utmost importance. The performance being measured primarily in terms of the specific thrust $(F/\dot{m}_0)$ and specific fuel consumption (S.F.C.). These two parameters depend on the operating envelop of the aircraft and component performance of the engine. Attempts have been made by different authors [1,2,3] to evaluate the aero engine performance. Most of these methods estimate the engine performance assuming that the component efficiencies do not change with off-design conditions. It is also assumed that the gas constant remains constant during the evaluation of engine performance. Hence it is very important to estimate these parameters at different operating points of the aero engine by considering the above effects. For the this purpose a computer model was developed. In this model the variable component characteristics and specific heats of the gas is built in as separate modules. The model also provides for correction in mass flow with bleed air, cooling air flow and fuel-air ratio [3]. The developed software model was used to design and predict the performance of a typical twin-spool turbojet engine at different flight conditions. The effect on engine performance due to variation in component efficiencies and specific heats as against constant efficiency and specific heat has been compared and the results are presented.

Nomenclature

a	: acoustic velocity (m /sec)
e	: polytropic efficiency
f	: fuel-air ratio
h	: calorific value of fuel (kJ/kg)
hp	: high pressure spool
lp	: low pressure spool
$\dot{m}$	: mass flow rate (kg/sec)
u	: velocity (m/sec)
A	: area (m^2)
Cp	: specific heat coefficient at constant pressure (kJ/kg K)
M	: Mach number
P	: static pressure (kPa)
P_t	: total pressure (kPa)
R	: universal gas constant (kJ/kg K)
T	: static temperature (K)
T_t	: total temperature (K)
$F/\dot{m}_0$	: specific thrust (N/sec)
S.F.C.	: specific fuel consumption (mg-sec/N)
π	: total-total pressure ratio
τ	: total-total temperature ratio
η	: isentropic efficiency
β	: bleed air flow fraction
ε	: cooling air flow fraction
γ	: ratio of specific heats

Subscripts

b	: burner
c	: compressor
cp	: lp-compressor
ch	: hp-compressor
d	: diffuser
est	: estimated
in	: initial
m_1	: hp-spool mechanical
m_2	: lp-spool mechanical
n	: nozzle

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r : ram
ref : reference
t : turbine
th : hp-turbine
tl : lp-turbine

Note

$$\tau_\lambda = \frac{C_{pt}T_{t4}}{C_{pc}T_0}$$

$$\tau_r = 1+\frac{\gamma-1}{2}M_0^2 = \frac{T_{t0}}{T_0}$$

$$\pi_r = \left(1+\frac{\gamma-1}{2}M_0^2\right)^{\gamma/\gamma-1} = \frac{P_{t0}}{P_0}$$

2. CYCLE ANALYSIS

The engine cycle analysis was carried out at sea-level static as a design case. The results obtained from this design case were used for off-design analysis. The station convention and the schematic layout of a typical aero engine is shown in Figure 1.

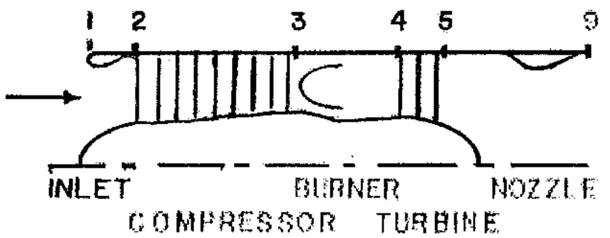

Fig.1 Engine layout and station convention

2.1 Design Point Analysis

The specific thrust $(F/\dot{m}_0)$ for a turbojet engine is given by,

$$F/\dot{m}_0 = a_0\left\{Y*\left(M_0\frac{u_9}{u_0}\right) - M_0 + \frac{Y*\left(\frac{T_9}{T_0}\right)}{\gamma_c\left(M_0\frac{u_9}{u_0}\right)}\left(1-\frac{P_0}{P_9}\right)\right\} \quad (1)$$

where $y = \dot{m}_9 / \dot{m}_0$

We see from the above equation to estimate the specific thrust we require to estimate the terms (T_9/T_0) and (u_9/u_0).

$$\frac{T_9}{T_0} = \frac{C_{pc}}{C_{pt}} * \frac{\tau_{th} * \tau_{tl} * \tau_\lambda}{\left(\pi_r\pi_d\pi_{cp}\pi_{ch}\pi_b\pi_{th}\pi_{tl}\pi_n\frac{P_0}{P_9}\right)^{(\gamma_t-1)/\gamma_t}}$$

The term (u_9/u_0) is given by,

$$\left(\frac{u_9}{u_0}\right)^2 = \left(\frac{M_9}{M_0}\right)^2 * \left(\frac{\gamma_9 R_9}{\gamma_0 R_0}\right) * \left(\frac{T_9}{T_0}\right) \quad (3)$$

The specific fuel consumption (S.F.C.) is given by,

$$S.F.C. = \frac{f}{F/\dot{m}_0} \quad (4)$$

Where f is the fuel-air ratio which is estimated from [3] as follows,

$$f = 0.99 * (T_{t3} - T_{t2} - 10) * \left(1+\frac{T_{t2}}{3250}\right) \Big/ (h * \eta_b) \quad (5)$$

for $400 > T_{t3} - T_{t2} > 200$

and,

$$f = 1.1 * (T_{t3} - T_{t2} - 50) * \left(1+\frac{T_{t2}}{3250}\right) \Big/ (h * \eta_b) \quad (6)$$

for $900 > T_{t3} - T_{t2} > 400$

2.2 Off-Design Analysis

The off-design analysis was carried out in two modes. In the first mode it is assumed that the hp-turbine is choked and the hp-turbine pressure ratio is equal to the hp-turbine reference pressure ratio. This would result in the operating point on the hp-turbine getting fixed at a single point. The area of the nozzle is also assumed to be fixed, this would more or less fix the operating point on the lp-turbine. Depending on the inlet conditions to the compressor the operating point gets slightly altered. Indicating a small change in compressor efficiency, which at present is neglected. The governing equations for this mode of analysis is explained in Section 2.2.1.

In the second mode it is assumed that hp-turbine is choked, hence the mass flow parameter through the hp-turbine is fixed. The pressure ratio, efficiency of the hp and lp turbine were varied to match with the hp and lp compressor. Here at the off-design point the operating point on the turbine and compressor characteristics will be different than the design reference point. The iterative procedure used for this mode of analysis is explained in Section 2.2.2.

Mode 1 : Fixed Operating Point On Hp-turbine

A fixed area turbojet is considered. In which the hp-turbine inlet nozzle is assumed to be choked at all flight operating conditions. For a fixed area turbojet we can show that [1,4],

$$\frac{\tau_t^{1/2}}{\pi_t} = \frac{A_8}{A_4} \tag{7}$$

Using the above equation and with the assumption that the area A_4 is a constant at all operating points of the engine, we arrive at the following relationship for the pressure-ratio across the lp-turbine,

$$\pi_{tl} = \pi_{tl,ref} * \frac{A_{8,ref}}{A_8} * \left(\frac{\tau_{tl}}{\tau_{tl,ref}}\right)^{1/2} \tag{8}$$

The temperature ratio τ_{th} is obtained from the power balance between turbine & compressor on the hp-spool and is given by,

$$\tau_{th} = 1 - \frac{\tau_d \tau_r}{\eta_{ml} \tau_\lambda} \frac{(\tau_{cp}\tau_{ch} - \tau_{cp})}{(1-\varepsilon_1-\varepsilon_2-\beta+f)} \tag{9}$$

Since the hp-turbine is choked $\pi_{th} = \pi_{th,ref}$.

Using the above relation and the reference conditions, the temperature ratio across lp and hp compressors are given by,

$$\tau_{cp} = 1 + (\tau_{cp,ref} - 1) * \frac{(\tau_\lambda/\tau_r\tau_d)}{(\tau_\lambda/\tau_r\tau_d)_{ref}} * \frac{(1-\tau_{tl})}{(1-\tau_{tl})_{ref}} * \frac{(1-\varepsilon_1-\varepsilon_2-\beta+f)}{(1-\varepsilon_1-\varepsilon_2-\beta+f)_{ref}} \tag{10}$$

$$\tau_{ch} = 1 + (\tau_{ch,ref} - 1) * \frac{(\tau_\lambda/\tau_r\tau_d\tau_{cp})}{(\tau_\lambda/\tau_r\tau_d\tau_{cp})_{ref}} * \frac{(1-\varepsilon_1-\varepsilon_2-\beta+f)}{(1-\varepsilon_1-\varepsilon_2-\beta+f)_{ref}} \tag{11}$$

The mass flow rate at the inlet of the engine is given by,

$$\dot{m}_0 = \left(\frac{\Gamma}{\sqrt{R}}\sqrt{\frac{C_{pt}}{C_{pc}}}\right)\pi_r\pi_d\pi_{cp}\pi_{ch}\pi_b \frac{P_0}{\sqrt{T_0}} * \frac{A_4}{\sqrt{\tau_\lambda}} \tag{12}$$

where $\Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/(2*(\gamma-1))}$

The equations (8), (9), (10), (11) and (12) are used in an iterative manner to evaluate the off-design performance of the engine.

2.2.2 Mode 2 : Variable Operating Point On Hp-Turbine

The steady-state engine performance at each speed is determined by two conditions: mass flow balance and a power balance. The turbine mass flow must be sum of the compressor mass flow and fuel mass flow, minus compressor bleed and cooling air flow. Also the power output of the turbine must be equal to that demanded by the compressor.

While component matching it is assumed that :-

a) Temperature at inlet to hp-turbine (T_{t4}) remains constant.

b) The hp-turbine remains choked at all flight conditions.

c) Non-dimensional compressor shaft speed = Non-dimensional turbine shaft speed.

The following procedure was used in matching the components of the engine:

Initially an arbitrary $(\dot{m}\sqrt{T}/P)$ for the given speed parameter $(N/\sqrt{T})$ at the inlet of the lp-compressor was chosen. Using the component performance map, the corresponding pressure ratio developed across the lp-compressor and its efficiency were evaluated. Then the work done by the lp-compressor was calculated using the pressure ratio, efficiency and the inlet conditions (i.e., temperature & pressure). An arbitrary pressure ratio across hp-turbine corresponding to the operating condition of the hp-turbine, being choked was selected. Using the pressure ratio across the lp-compressor, the choked mass flow parameter and the pressure drop across the burner, the pressure ratio developed across the hp-compressor was calculated. Using the pressure ratio across the hp-turbine, the efficiency $(\eta_{th,in})$ was estimated. The pressure ratios developed across lp and hp compressor's were used to calculate the fuel-air ratio. From the power balance for the hp-spool, using the hp-compressor performance characteristics a new value of the hp-turbine efficiency $(\eta_{th,est})$ was estimated. In case the difference $(\eta_{th,est} - \eta_{th,in})$ was not negligible, then the pressure ratio across the hp-turbine was incremented and the procedure repeated from starting to the end. With the hp-spool matched, only the components on the lp-spool were to be matched. For this the results of the previous analysis were used and the inlet conditions of the lp-turbine were calculated. Using the lp-turbine performance characteristics and the known inlet conditions (i.e., mass flow parameter) the corresponding pressure ratio across the lp-turbine and also its efficiency were obtained. Finally the work done by the lp-turbine was calculated. The matching procedure was repeated until the work done by the lp-turbine is equal to that of the lp-compressor by varying the mass flow parameter of the lp-compressor.

3. ANALYSIS

For analysis a twin-spool turbojet engine was chosen with a cycle pressure ratio of 7, this would give $\pi_{ch} = \pi_{cp} = 2.646$. The temperature at the inlet of hp-turbine $\left(T_{t4}\right)$ was fixed at 1295 K due to design limitations. To account for frictional losses in fluid flow a pressure drop of 5% across the diffuser, 3% across the burner and 2% across the nozzle were considered. A loss of 2% in efficiency of the burner and 5% in the mechanical shaft efficiency of the lp and hp spool were also considered. The polytropic efficiencies of the lp and hp compressor's and turbines were assumed to be, 0.83 and 0.84. For complete expansion of the gas at the nozzle exit the ratio of static pressure's (P_0 / P_9) was taken as unity and appropriate cooling air flow (1%) and bleed mass flow rate (1%) were also included in the analysis.

The specific heats used depend on the compression and expansion processes. Standard graphs which give the variation of specific heat with temperature after compression and variation of specific heats with the turbine inlet temperature were first digitized and then computerized in the form of polynomial equations. These equations were used to estimate the mean specific heats for each component.

For component matching the turbojet engine was evaluated at design speed$\left(N / \sqrt{T} = 1.0\right)$ operation for both lp and hp shafts. The performance characteristics of the compressor's and turbines are described by polynomial equations of order 3. These equations give the variation of pressure-ratio and component efficiency against non-dimensional mass flow rate for various non-dimensional speed's. The ambient pressure, temperature and density were corrected for change in altitude. The performance of the turbojet configuration was estimated for the flight Mach numbers, varying from 0 to 1.0, at two different flight altitudes of 0 km and 10 km. The evaluation was carried out under the following headings:

(a) Constant specific heat coefficients and co... efficiencies. This is considered as reference case ... future discussions.

(b) Variable specific heat coefficients but constant component efficiencies.

(c) Variable specific heat coefficients and component efficiencies.

The off-design analysis was carried out with sea-level static performance as the reference. For the above analysis the variation in specific thrust and specific fuel consumption were estimated and represented graphically as variation of these parameters with flight Mach number for a given altitude.

4. RESULTS AND DISCUSSIONS

Figure 2 represents the variation in performance parameters at a flight altitude of 0 km. In this figure the variation of specific thrust and specific fuel consumption has been plotted against flight Mach number for a given altitude. It is observed from this figure that at sea-level static condition $\left(M_0 = 0.01\right)$ the magnitude of specific thrust is highest with variation in specific heat and efficiency as compared to reference case (a). There is a marginal difference in specific thrust with and without efficiency correction. But as forward velocity increases it is observed that specific thrust for case (c) increases for a flight Mach number range of, 0.2 - 0.8, this rise is significant and higher than case (b). With a further increase in flight Mach number the specific thrust does not change much with variation in specific heat and efficiency. This is clearly noticed at a flight Mach number of unity.

The variation of S.F.C. with flight Mach number for the three different cases is also shown in figure 2. We observe from this figure that the effect of efficiency on specific fuel consumption (S.F.C.) is predominant as compared to the effect of specific heat. It is also observed that the S.F.C. is always higher than the reference case (a) for the range of flight Mach numbers considered. This is because while evaluating the off-design performance with component efficiencies, the component's efficiency values turned out to be lower than the reference case (a). The specific heat variation is observed to have a small effect on S.F.C. The effect of variation in specific heats is to reduce S.F.C. as compared to reference case (a) for all flight Mach number. This is because the value of specific heat corresponding to the temperatures at inlet and outlet of combustion chamber are lower than the reference case and the effective enthalpy rise across the combustion chamber is lower than the reference case (a).

Figure 3, shows the variation in performance at 10 km flight altitude. The specific thrust for the three cases is clearly distinguishable for the flight Mach number range, 0.01 - 0.4. With a further increase in the flight Mach number specific thrust for variation in specific heat and efficiency becomes higher than the reference (case a), the difference is only marginal. Thus on comparing figures 2 and 3 it is seen that the effect of specific heat on the engine performance depends on altitude. The variation in S.F.C. follows the same pattern as that at a flight altitude of 0 km, with the exception that the difference in S.F.C. of case (a) and (b) becomes more significant.

The variation in performance parameter is better understood by analyzing the percentage variation in component performance at the two altitudes. These variations are defined as follows:
Variation(%) in F/m or S.F.C.

$$= \frac{\text{value of case (b) or (c) - value of case (a)}}{\text{value of case (a)}}$$

The performance parameters like specific thrust and specific fuel consumption are inserted for the value of the cases. The percentage variation of specific thrust and S.F.C. with flight Mach number for the two altitudes of 0 and 10 km are shown in figures 4 and 5. Consider first the variation at 0 km. It is clear that for the range of flight Mach number , 0.2-1.0, there is not much effect of specific heat correction on the percentage variation of specific thrust as well as S.F.C. But the effect of correction in component efficiency on the variation of specific thrust is significant with a value as high as 2% at $(M_0 = 0.6)$ which then drops to 0.30 %, with further increase in flight Mach number. The percentage variation in the S.F.C. due to efficiency correction is seen to drop from 5% to 3% and then again rises with increase in flight Mach number.

The percentage variation of component performance (fig.5) at 10 km altitude indicate that the effect of efficiency correction dominates the variation of specific thrust and S.F.C. whereas effect due to specific heat correction is observed to be negligible. The percentage variation of specific thrust with efficiency correction rises continuously from -1.25% to +1.25% for an increasing forward speed. The percentage variation in S.F.C. is observed to drop from 11% to 6% with higher flight Mach numbers.

From figures 4 and 5, it is be concluded that the effect of component efficiency correction on the specific fuel consumption is predominant at higher altitudes. Also the graphs representing the percentage variation clearly define that the correction in efficiency of the components is the main cause for the shifts in specific thrust and S.F.C. of the turbojet observed in figures 2 and 3 at different flight conditions.

5. SENSITIVITY OF ENGINE PERFORMANCE WITH COMPONENT EFFICIENCIES

The earlier analysis involved variation in efficiencies of both components like, compressor and turbine. Further analysis was carried out to find the effect of individual component efficiency on engine performance parameters like specific thrust and specific fuel consumption. The calculations were carried out to estimate specific thrust and S.F.C. for a unit change in a particular component efficiency while maintaining the other component efficiencies same. It is assumed that each stage of lp and hp compressor/turbine have same efficiencies. It is observed from Table I, that there is a 1.2% increase in specific thrust and a decrease of 0.6% in S.F.C. for 1% change in the polytropic efficiency of both lp and hp compressors. Whereas there is only a 0.64% increase in specific thrust and a decrease of 0.6% in S.F.C. for 1% change in polytropic efficiency of both lp and hp turbine. This result clearly shows that the change in efficiency of the lp and hp compressors contribute in a larger magnitude to the change in specific thrust and S.F.C.

6. CONCLUSION

The aero-engine performance is primarily dependent on the operating efficiencies of the various components, especially the compressor which plays the major role. The real gas effects are important in the evaluation of aero-engine performance since the properties of the working fluid are dependent on its composition and the local temperature and pressure. The effect of variation in component efficiency on the specific thrust is predominant at Mach numbers lower than unity at lower altitudes and at Mach numbers greater than unity at higher altitudes. The specific fuel consumption also depends on the component efficiency at all flight altitudes and flight Mach numbers. The effect of component efficiency on the specific fuel consumption is more predominant at higher altitudes.

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8. REFERENCES

1. Oates, G.C., "Aerothermodynamic of Gas Turbine and Rocket Propulsion", American Institute of Aeronautics and Astronautics, Inc., New York, 1984.

2. Shepherd, D.G., "Aerospace Propulsion ", American Elsevier Publishing Company, Inc., New York, 1972.

3. Harman, Richard T.C., "Gas Turbine Engineering", The Macmillan Press Ltd., Hong Kong, 1981.

4. Katari, Ashutosh, "Design and Off-Design Performance Evaluation of a Aero Engine", B.E.(Hon's) Mechanical Degree (BITS, PILANI) Project Report, National Aerospace Laboratories, Bangalore, June 1997

TABLE I : Sensitivity of engine performance with component efficiencies:

Cases	η_c		η_t		Change in $F/\dot{m}_0$ (%)	Change in S.F.C (%)
Change in compressor polytropic efficiency	0.83	0.84	0.84	0.84	1.205	-0.673
Change in turbine polytropic efficiency	0.83	0.83	0.84	0.85	0.648	-0.643

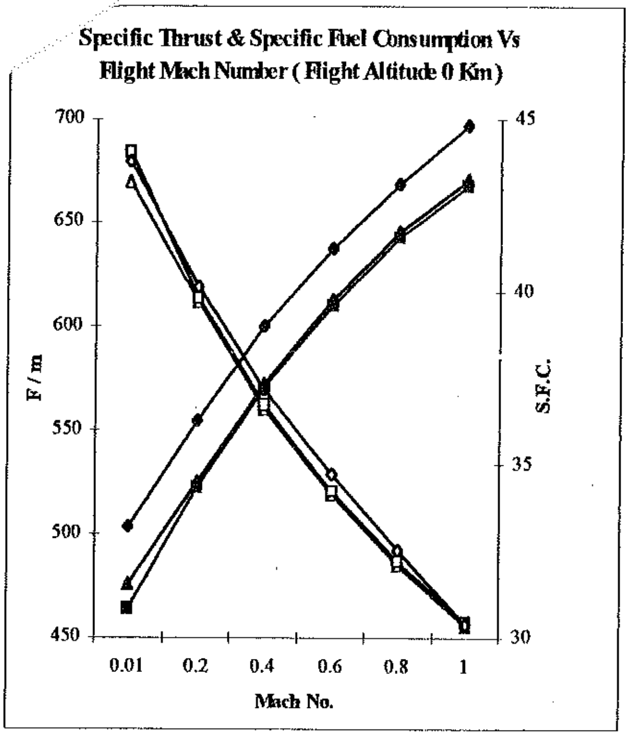

Fig. 2

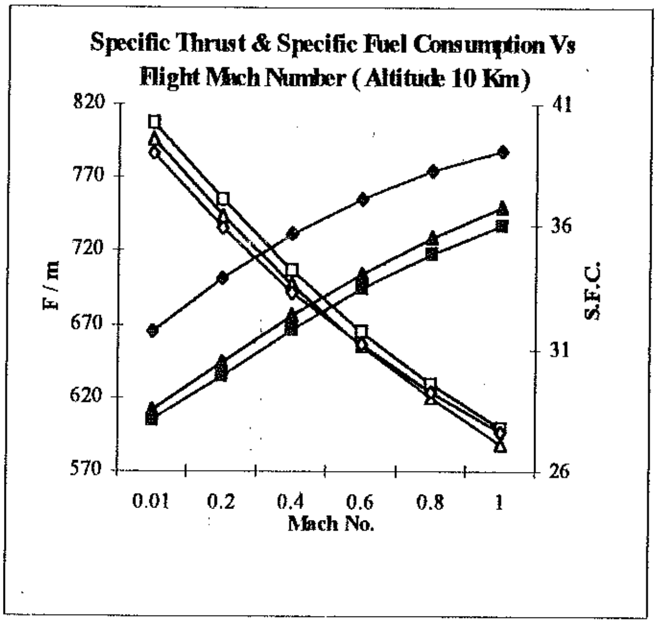

Fig. 3

F / M	S.F.C.	
△	▲	With No Cp Correction and no matching
□	■	With Cp Correction and no matching
◇	◆	With Cp & Efficiency Correction and matching

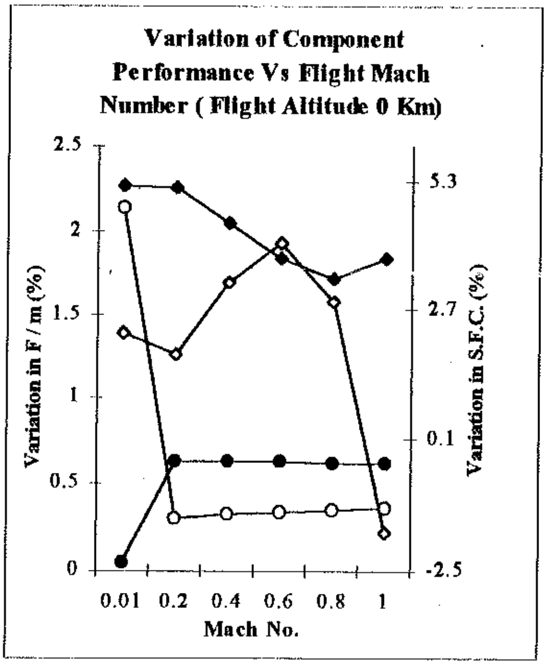

Fig. 4

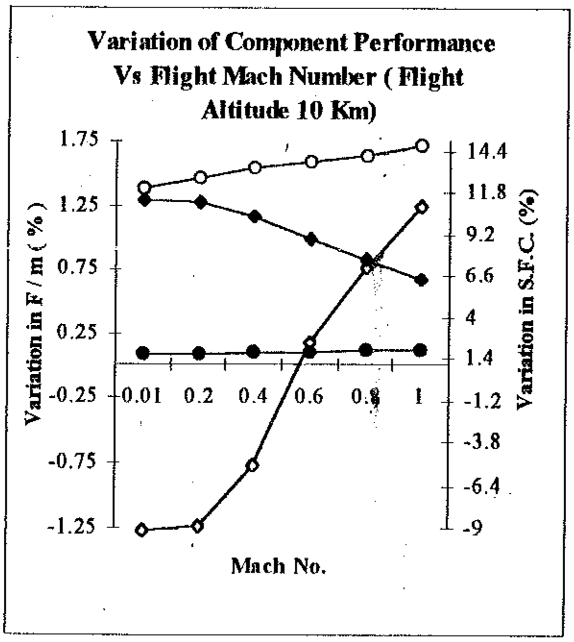

Fig. 5

F / M	S.F.C.	
○	●	With Cp Correction and no matching
◇	◆	With Cp & Efficiency Correction and matching